

# A Comparison of Nuclear Thermal and Nuclear Electric Propulsion for Interplanetary Missions

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A comparison is made between nuclear thermal rockets and nuclear powered electric propulsion systems. Complete missions are designed to be launched by a single Ariane 5 and fly by Jupiter and Pluto powered either a nuclear thermal system or a nuclear powered electric system. It is shown that it is feasible to build both nuclear thermal and nuclear electric missions to Jupiter and Pluto. Nuclear thermal systems are designed to go to Jupiter with an inert mass fraction of 0.6094, a power of 281.23 MW and a flight time of 4.13 years. Nuclear thermal systems are designed to go to Pluto with an inert mass fraction of 0.4182, a power of 281.23 MW and a flight time of 19 years. Nuclear electric systems are designed to go to Jupiter with an inert mass fraction of 0.5266, a power of 10.258 kW and a flight time of 4.13 years. Nuclear electric systems are designed to go to Pluto with an inert mass fraction of 0.2656, a power of 10.258 kW and a flight time of 19 years. The results of the system designs are analyzed and an outlook is given for each system. Additional issues of testing, safety and radiological hazards are discussed.

## Nomenclature

$a$	=	ion thruster sizing constant	$m_t$	=	thruster mass
$b$	=	ion thruster sizing constant	$m_{tank}$	=	tank mass
$\vec{F}$	=	thrust	$P$	=	power
$F/W$	=	thrust to weight ratio	$p_b$	=	burst pressure
$f_{inert}$	=	inert mass fraction	$P_{core}$	=	power of the reactor core
$g_0$	=	earth's gravitational acceleration	$R_{core}$	=	radius of the reactor core
$H_{core}$	=	height of the reactor core	$T$	=	temperature of the reactor
$I_{sp}$	=	specific impulse	core		
$\dot{m}$	=	mass flow rate	$TOF$	=	time of flight
$m_f$	=	final mass	$\Delta V$	=	change in velocity
$m_{Hall}$	=	mass of a Hall thruster	$V_e$	=	exhaust velocity
$m_i$	=	initial mass	$V_{tot}$	=	total propellant volume
$m_{pay}$	=	payload mass	$\varphi_{tank}$	=	tank mass factor
$m_{prop}$	=	propellant mass	$\eta$	=	total efficiency

## Introduction

In the infancy of mankind's great venture into space we have already discovered some of the limitations of chemical propulsion. It has been the mainstay of our propulsion technology, lifting payloads into orbit and sending them to the far reaches of our solar system, and even beyond. However, chemical propulsion has limits to its efficiency and its energy density. New propulsion technology promises to be the next stepping stone into space. Nuclear technology allows an extended reach for spacecraft to get to the outer planets and even beyond our solar system. This study attempts to prove the feasibility of nuclear propulsion systems by designing and comparing

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spacecraft with nuclear thermal and nuclear powered electric propulsion in the context of missions to Jupiter and Pluto.

There are three different ways to use nuclear power: in a Nuclear Thermal Rocket (NTR), as Nuclear Electric Propulsion (NEP) or as a Radioisotope Thermoelectric Generator (RTG). NTR systems use the thermal energy of a nuclear reactor to heat a propellant, usually hydrogen, to high temperatures and then release them as exhaust. Because of the very high temperatures achieved by fissioning radioactive material, this system creates a large thrust at a specific impulse much higher than that of chemical rockets. NEP systems use a nuclear reactor to generate electricity to then power an electric thruster, such as an ion thruster or Hall thruster. These low-thrust propulsion systems are extremely efficient. An alternate way to create lower levels of electricity to power a thruster is to use an RTG, which converts the heat of decaying radioactive material into electricity.

Nuclear power in space is not a new concept. The 1950's saw the creation of Project Rover initially located in Los Alamos, New Mexico, which sought to develop nuclear thermal rocketry and supported one program in particular, the Nuclear Engine for Rocket Vehicle Applications (NERVA). The NERVA program achieved significant results, including 28 full power tests, 890,000 N thrust and reactors up to 200,000 MW.<sup>1</sup> Approximately 300 graphite and uranium-carbide fuel elements heated the liquid hydrogen propellant. Funding was cut for NERVA in the early 1970's, but it remained the most significant nuclear thermal rocket test program performed to date.

The Space Nuclear Auxiliary Power (SNAP) program of the 1960's developed the RTG as a viable space power option. Almost all of the nuclear systems that have flown in space have been RTGs, and these systems have been shown to be reliable and stable over the long term.

The late 20<sup>th</sup> century saw the development of the Particle Bed Reactor (PBR). Instead of fuel elements, tiny graphite and zirconium carbide coated spheres of uranium carbide are directly cooled by hydrogen propellant. It was developed by the Air Force's Space Nuclear Thermal Propulsion (SNTP) program, which ended in 1993. The PRB is promising because of its high specific impulse, thrust and thrust-to-weight ratio.

There are a variety of electric thrusters that can be powered by a nuclear reactor or an RTG. Ion engines and Hall thrusters were considered for this study. Magnetoplasmadynamic (MPD) thrusters were considered as another alternative because of their high specific impulse and relatively high thrust, but were ultimately rejected as being too experimental at this point in time. MPD's are only efficient with megawatt levels of power, which is beyond the scope of the Project Prometheus thrusters, which operate up to 100 kWe.

Gridded Ion thrusters operate on the principal of ionizing a propellant, normally xenon, and accelerating it through electrically charged grids in order to produce a thrust. Figures 1 and 2 show schematics and pictures of ion thrusters. Several variations exist, differing by the method of ionizing the propellant. The Kaufman-type thrusters work by ionizing the propellant in a cylinder with an electric field and a magnetic field. It was one of the original types of ion thrusters. The SERT 2 used this type of thruster and was the first gridded thruster used in space. Current examples of this system include the British T5 and T6. Magneto-Electrostatic Containment (MESC) thrusters are very similar to Kaufman thrusters, except that they operate with much stronger magnetic fields. Examples of this include XIPS-13, XIPS-25, and the NSTAR, which flew on the Deep Space 1 spacecraft. The third major type of ion thruster is the Radiofrequency Ionization Thruster (RIT). It uses a high frequency oscillating current that is passed through a coil wrapped around the discharge chamber, creating an alternating axial magnetic field, and an azimuthal electric field. The German RIT-10, 35 and XT are all RIT thrusters.

Hall thrusters are a work by using axial electric fields and radial magnetic fields to create a "Hall" current in plasma acting in an azimuthal direction. The current, in turn, reacts with a magnetic field to create a downstream axial force. A schematic is presented in Figure 3. The large

ion current density allows a greater thrust than ion engines. This higher thrust and high specific impulse has caused several agencies, such as the Department of Defense, to invest significant money into Hall thruster development.

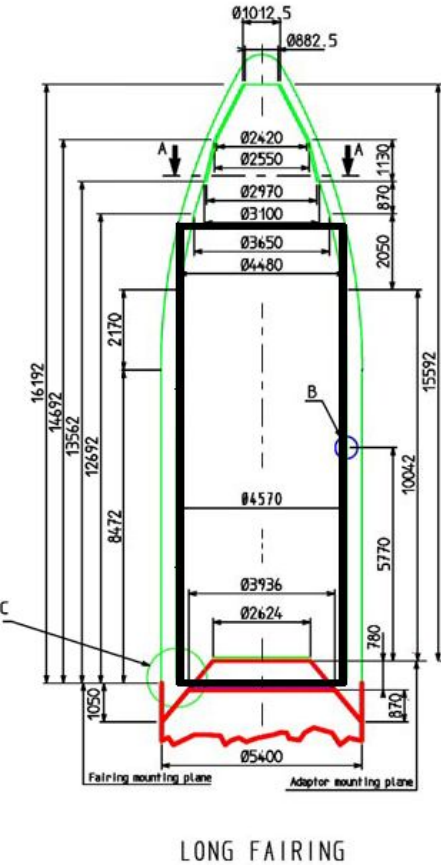
Project Prometheus is a current NASA program to investigate NEP and develop the associated technologies. The Jupiter Icy Moons Orbiter (JIMO) is slated to be the first spacecraft with Prometheus propulsion on board. With all of the current emphasis on electric propulsion, this study attempts to look at both nuclear thermal and nuclear electric propulsion and design feasible missions to go to Jupiter and Pluto. This study will also address the issues surrounding nuclear propulsion, including testing, safety and radiological hazards.

### Preliminary System Sizing

The first level of analysis sought to determine acceptable inert mass fractions based on a given initial mass and an estimate of the specific impulse for given systems. The  $\Delta V$ 's for the missions are given in Table 1.

**TABLE 1: Mission design  $\Delta V$ 's<sup>2</sup>**

	NTR $\Delta V$ (km/sec)	NEP $\Delta V$ (km/sec)	NTR TOF (years)	NEP TOF (years)
Jupiter Fly-by	3.83	7.66	4.13	4.13
Pluto Fly-by	6.70	13.40	19.00	19.00



**Figure 4: Ariane 5 Long Fairing with volume approximation (mm)**

approximately half the  $\Delta V$  of NEP system with the same time of flight.

The NEP numbers are based on an Earth-Venus-Earth-Jupiter gravity assist and an Earth-Venus-Jupiter-Pluto gravity assist. The NEP  $\Delta V$ 's had been specifically developed for low thrust missions to fly by Jupiter and Pluto by Deban, McConaghy and Longuski of Purdue University, and were incorporated into this study as one of the few cases where  $\Delta V$ 's for low-thrust trajectories had been calculated. In order to compare the NEP missions to a similar NTR mission and to incorporate the lack of NTR gravity losses for due to impulsive burns, the NTR  $\Delta V$ 's were assumed to be 50% of the NEP  $\Delta V$ 's. This has been shown historically to be a valid approximation. To find approximations for the NTR time of flight, a study by Zubrin and Sulmeisters was used, which calculated trip times and  $\Delta V$ 's for a mission to Pluto. The Zubrin/Sulmeisters and Deban/McConaghy/Longuski numbers are compared in Table 2.

**TABLE 2: Time of flight comparison<sup>3</sup>**

	Mission	$\Delta V$ (km/s)	TOF (yrs)
Deban/McConaghy/Longuski	Pluto NEP	13.4	19
Zubrin/Sulmeisters	Pluto NTR	6.52	16
Zubrin/Sulmeisters	Pluto NTR	12.9	10

This table shows that for an NTR mission, the time of flight is roughly half of the time of flight of an NEP mission with the same  $\Delta V$ . This validates the approximation that NTR's have approximately half the  $\Delta V$  of NEP system with the same time of flight.

In order to meet the design criteria that the spacecraft be launched on one launch vehicle, the entire structure must fit on a single launch vehicle. The Ariane 5 was chosen as the design launch platform because it is one of the largest available. Its lift capacities are shown in Table 3. The Long Fairing is approximated as a cylinder, shown in Figure 4.

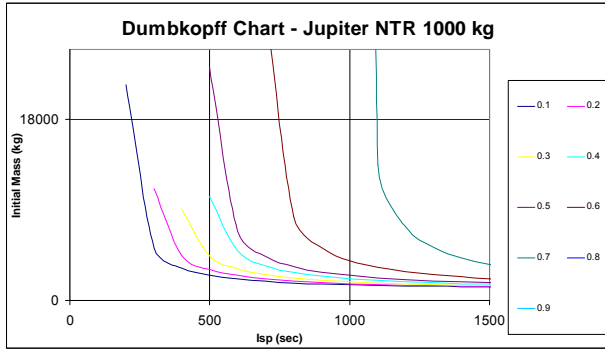
**TABLE 3: Ariane 5 Payload Specifications**

Mass to orbit (kg)	18000
Height (m)	12.5
Diameter (m)	4.5

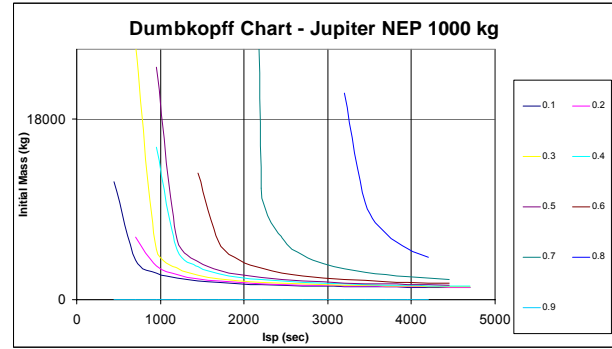
With this data as starting points, the preliminary system specifications can be established using Dumbkopff charts. There is a relationship between initial mass, payload mass,  $\Delta V$ , specific impulse and the inert mass fraction given in equation 1 that governs these charts.

$$m_i = \frac{m_{pay}(1 - f_{inert})}{e^{\left(\frac{-\Delta V}{I_{sp}g_0}\right)} - f_{inert}} \quad (1)$$

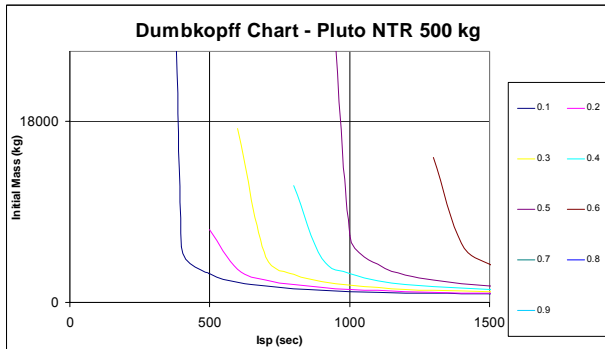
Graphs were created for a 1000 kg payload to Jupiter and a 500 kg payload to Pluto. A mission of 500 kg was determined to be on the lower end of system masses that could still accomplish useful science. Higher mass could be sent to Jupiter because of the lower  $\Delta V$ . Figures 5-8 show these system sizing charts.



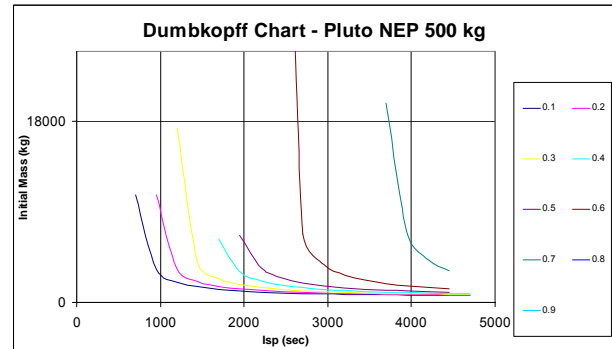
**FIGURE 5: 1000 kg Jupiter NTR mission**



**FIGURE 6: 1000 kg Jupiter NEP mission**



**FIGURE 7: 500 kg Pluto NTR mission**



**FIGURE 8: 500 kg Pluto NEP mission**

These graph yield starting points for the design of the two spacecraft to each Jupiter and Pluto. These operating points are listed in Table 4.

**TABLE 4: Design points for system design**

	Design Isp (sec)	$\Delta V$ (km/sec)	f-inert
Jupiter NTR	1000	3.83	0.65
Jupiter NEP (Ion)	3500	7.66	0.80
Jupiter NEP (Hall)	1500	7.66	0.60
Pluto NTR	1000	6.70	0.50
Pluto NEP (Ion)	3500	13.40	0.65
Pluto NEP (Hall)	1500	13.40	0.32

## NTR Design

The methodology outlined in *Space Propulsion Analysis and Design* is used to design the nuclear thermal rocket. The goal is to take a series of inputs –  $\Delta V$ , inert mass fraction, payload mass, payload sizing limitations – and use a series of calculations and reasonable design assumptions to size a rocket capable of reaching the given  $\Delta V$ .  $\Delta V$  is used as an input into the equation, but due to the approximate and empirically-based design equations, it must be recalculated once the system is sized in order to ensure that the system will reach the required  $\Delta V$ . An NTR system is considered viable if it can reach the given  $\Delta V$ , fit within the payload fairing, and have an initial mass less than the lift capabilities of the Ariane 5 (see Table 2).

Some preliminary parameters must be defined before the subsystems can be designed. Initial assumptions include a specific impulse of 1000 seconds, which assumes hydrogen propellant, 7 MPa chamber pressure, and a core temperature of 3200 K. These parameters are all easily achievable by today's nuclear technology. The initial thrust to weight ratio was held at .3 through trial and error; this turned out to be a good balance between a high thrust short burn to approximate an impulsive burn and a low enough thrust to keep the reactor small. The following equations establish some basic parameters of the system:

$$m_{prop} = \frac{m_{pay} \left( e^{\left( \frac{\Delta V}{I_{sp} g_0} \right)} - 1 \right) (1 - f_{inert})}{1 - f_{inert} e^{\left( \frac{\Delta V}{I_{sp} g_0} \right)}} \quad (2)$$

$$\vec{F} = \frac{F}{W} m_i \quad (3)$$

$$\dot{m} = \frac{\vec{F}}{I_{sp} g_0} \quad (4)$$

Using the mass flow rate and the total propellant mass, burn time can be calculated. Since the  $\Delta V$ 's are assumed to have impulsive burns, the burn time must be small compared to the trip time. For this modeling process, if the burn time was greater than 1% of the trip time, the system specifications were reevaluated. The reactor power needed to create this thrust can be found using an equation that determines the power required to heat up 1 kg/s of a given propellant to a given temperature. Equation 5 gives this relationship for hydrogen gas, and is accurate to several percent.

$$P = \dot{m}(0.018061T - 5.715417) \quad (5)$$

With these preliminary requirements established, the subsystems could be sized appropriately.

The payload mass was established at 1000 kg to Jupiter and 500 kg to Pluto at the beginning of the project. In order to determine an appropriate volume for the payload, several real space

missions were investigated to discover the relationship between linear dimensions and mass. The payload densities varied, from 367 kg/m<sup>3</sup> for ESA's SMART-1 to 30 kg/m<sup>3</sup> for the Cassini mission to Saturn. For this project, the payload was assumed to be a cylinder with a density of 100 kg/m<sup>3</sup> and given a height of 2 m, and the radius sized based on the given density and height, making the diameter slightly longer than the height.

The propellant tank tended to be a limiting factor, not because of its mass, but because of its size. Hydrogen stores cryogenically at a density of only 71 kg/m<sup>3</sup>, which means there must be a large volume available to accommodate the propellant. To size the tank, it was assumed to be a cylindrical tank in order to maximize volume usage in the prolate payload fairing. A spherical tank would have been better for surface area efficiency, but because the fairing height was so much greater than the diameter, cylindrical tanks were chosen. To calculate the mass of the tank, equation 6 was used

$$m_{\text{tank}} = \frac{p_b V_{\text{tot}}}{g_0 \phi_{\text{tank}}} \quad (6)$$

The tanks were designed as composite, so the tank mass factor was 10,000 m.

The turbopump feed system was designed based on a method that found the pressure at several points in the feed system, calculated the pump power, rotation rates and torques, and used this to define a mass for the turbo pump. The pump was given a height of .4 m.

The nuclear reactor was sized according to the required power. A 19 Element Particle Bed Reactor (PBR) was the most effective for the power range yielded by the modeling process. Equations 7 and 8 determine the reactor size based on the power requirements. These equations were developed for Los Alamos reactor by evaluating criticality and power in a simulation.

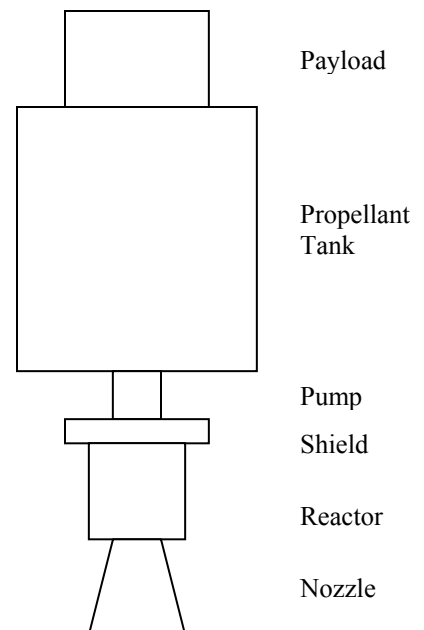
$$R_{\text{core}} = -2.655(10)^{-12} P_{\text{core}}^5 + 8.946(10)^{-9} P_{\text{core}}^4 - 1.1703(10)^{-5} P_{\text{core}}^3 + 7.427(10)^{-3} P_{\text{core}}^2 - 2.2955 P_{\text{core}} + 313.34 \quad (7)$$

$$H_{\text{core}} = -4.027(10)^{-5} P_{\text{core}}^2 + 0.1427 P_{\text{core}} + 17.9883 \quad (8)$$

The radiation shield was required to shield the payload from the damaging radiation emitted from the reactor. It is sized just slightly larger than the reactor and placed on the payload side of the reactor. A standard shield is sized for the reactor. This shield is made with 18 cm of beryllium, 5 cm of tungsten and 5 cm of lithium hydride. Each material is chosen for a reason: beryllium is a good neutron reflector, tungsten shields gamma rays and absorbs neutrons, and lithium hydride slows and absorbs neutrons. With this composition, the shield allows less than 0.2% of gamma rays through and neutrons are attenuated by 9 orders of magnitude.

The nozzle is designed to be ideally expanded in the vacuum of space. It is made from high strength columbium in order to withstand the temperatures of the exhaust propellant.

There are a number of other minor rocket subsystems required for operations that are placed in and around other components so that they are only factored into the design as



**FIGURE 9: NTR Layout**



mass considerations. They include: reactor containment vessel, cooling and feed system, attitude thrusters, structural mass and avionics. These components are sized according to the SPAD process, which is based on empirical data and usually sizes their mass as a fraction of another subsystem mass.

With all of the subsystems sized, the vehicle can then be analyzed again to determine if it will meet the three criteria to be a successful NTR. The mass criteria is simple to determine, because adding up all of the subsystem masses, the propellant mass and the payload mass gives the initial mass. As long as this is less than the lift capability of the Ariane 5, the design meets the mass criterion. To determine if the vehicle will fit in the fairing, the total rocket height must be determined. Each subsystem was already designed so that it met the width constraint of the fairing. The subsystems were stacked end on end in a standard configuration shown in Figure 9, and the total height was compared to the height of the payload fairing. Validating the  $\Delta V$  of the vehicle required the ideal rocket equation, shown as equation 9.

$$\Delta V = I_{sp} g_0 \ln \left( \frac{m_i}{m_f} \right) \quad (9)$$

Table 5 shows the results of the NTR design process.

**TABLE 5: NTR Design Results**

	$\Delta V$ (km/s)	f-inert	Initial Mass (kg)	Height (m)	Power (MWe)	TOF (years)
<b>Jupiter NTR</b>	4.191	0.6094	9100.41	7.23	281.23	4.13
<b>Pluto NTR</b>	8.103	0.4182	14853.83	12.29	281.23	19

## NEP Design

The design of the nuclear electric systems varied significantly from the design of the nuclear thermal rockets. This study incorporated several types of thrusters into the design, including gridded ion thruster variants such as the Kaufman thruster, the Magneto-Electrostatic Containment (MESC) thruster, and the Radio frequency Ionization Thruster (RIT), as well as the Hall thruster. The three types of ion thrusters all use electrified grids to accelerate ions, but vary in the methods of creating those ions from the xenon propellant. Validating NEP systems was limited to two criteria: meeting the initial mass specification of the Ariane 5 and meeting the  $\Delta V$  requirement for the mission. Preliminary analysis showed that the volume of the tank, the largest structure on the NEP vehicle was roughly one tenth the mass of an NTR tank for the same mission. The heavy components of the NEP system and the corresponding high inert mass fraction made the system fail for mass long before it would fail for volume. Therefore, volume analysis was neglected in the design of the NEP systems.

It is also necessary to comment on the power supply of the propulsion system. This study was intended to compare nuclear thermal and nuclear electric propulsion, however, the current standard mission power supply is solar arrays. Compared to nuclear power, they are cheap and free of the political, environmental and safety issues that surround nuclear power. The downside to solar power is that solar power density is inversely proportional to the square of the distance from the sun. Therefore, in order to draw the same amount of power as a 1 m<sup>2</sup> array in earth orbit, an array at Jupiter would have to be 27 m<sup>2</sup>, and an array at Pluto would have to be 1562 m<sup>2</sup>. This does not even account for degradation, which may be assumed to be 10% over the lifetime of the craft due to the interplanetary trajectory. The sheer size of the structure presents an almost insurmountable

engineering nightmare, even with the technology of thin film solar arrays. Masses for solar panels of this magnitude were calculated for the Jupiter missions, but thrown out on account of putting the designed spacecraft over the mass budget.

There are limits to the power range of nuclear reactors. The *SPAD* process sizes reactors for NTR systems to a lower limit of 100 MW, but critical reactors have been designed for as low a power as 6 kW. Below this power RTG's are used.

The NEP design process begins much like the NTR process: by assuming several values and then calculating basic design variables. Force, specific impulse and efficiency are all assumed based on the propulsion system. From there, propellant mass is calculated based on equation 2, mass flow rate with equation 4, and exhaust velocity and power are calculated with equations 10 and 11.

$$V_e = I_{sp} g_0 \quad (10)$$

$$P = \frac{1/2 \dot{m} V_e^2}{\eta} \quad (11)$$

Table 6 gives an idea of the current status of ion thruster research. The baselines used for this study will be the T6 for Kaufman thrusters, the XIPS-25 for MESC thrusters, and the RIT-XT for RIT thrusters. The table notation is as follows:

$D$	Nominal beam diameter (cm)	$\eta_e$	Electrical efficiency (%)
$F$	Nominal thrust (mN)	$\eta_t$	Total thruster efficiency (%)
$F(R)$	Demonstrated thrust range (mN)	$C_t$	Power/thrust ratio (W/mN)
$I_{sp}$	Specific impulse (sec)	$M_t$	Thruster mass (kg)
$P$	Total thruster input power (W)	$Stat$	Status: F = Flight qualified; UQ = Under qualification; EM = Engineering model
$\eta_m$	Propellant utilisation efficiency, including all losses (%)		

**TABLE 6: Models for ion thrusters<sup>4</sup>**

THRUSTER (Source)	Typical Reference	$D$ (cm)	$F$ (mN)	$F(R)$ (mN)	$I_{sp}$ (s)	$P$ (W)	$\eta_m$ (%)	$\eta_e$ (%)	$\eta$ (%)	$C_t$ (W/mN)	$M_t$ (kg)	Stat
<b>Kaufman</b>												
KERC 5 (Russia)	Gorshkov et al (1999)	5	3.9	1.5 – 6	2890	121	78	59	46	31.0	1.2	EM
T5 (UK)	Fearn and Smith (1998)	10	18	0.3 – 71	3248	463	79	78	62	25.7	1.6	F
Melco (Japan)	Ozaki et al (2000)	12	23.3	18.6 – 27.9	2897	616	80	78	62	26.4	4.9	U/Q
T6 (UK)	Wallace et al (1999)	22	150	30 – 200	3660	3870	86	82	70	25.7	6.5	EM
<b>MESC</b>												
XIPS-13 (USA)	Beattie et al (1993)	13	17.8	Set point	2585	439	79	69	54	24.6	5.0	F
XIPS-25 (USA)	Beattie et al (1985)	25	165	Set point	3800	4200	≈ 88	≈ 83	≈ 73	25.5	N/A	F
NSTAR (USA)	Christensen et al (1999)	30	92	19.5 – 92.7	3280	2310	84	85	71	25.1	8.3	F
<b>Radiofrequency</b>												
RIT-10 (Germany)	Killinger et al (2000)	10	15	5 – 25	3324	476	70	74	52	31.7	1.6	F
RIT-XT (Germany)	Leiter et al (2003)	21	100	15 – 185	4054	2570	93	83	77	25.7	N/A	EM
ESA-XX (ESA)	Bassner et al (1997)	25	200	10 – 240	3500	6800	85	85	72	34.0	N/A	EM
RIT-35 (Germany)	Groh et al (1990)	35	104	50 – 200	3193	2965	80	69	55	28.5	9.0	EM
<b>High Frequency RF</b>												
Muses-C (4.2 GHz) (Japan)	Kuninaka et al (2000)	12	8.1	4.3 – 8.1	2920	320	61	52	32	47.6	2.35	F
RMT (150 kHz) (Italy)	Capacci and Noci (1998)	9.5	5	2 – 8	3030	173	64	68	43	34.6	N/A	EM

As mentioned, the power system will either be an RTG or a nuclear reactor. Standard specific masses for these systems are given in Table 7. Note that the specific mass of the nuclear reactor does not include a radiation shield. The shield mass was calculated in the same manner as

for the NTR system. Solar specific masses are given for reference, having been previously rejected as a possible power source.

**TABLE 7: Specific masses of power systems**

	Specific Mass (kg/kW)
Solar (LEO)	10
Solar (Jupiter)	270.7
Solar (Pluto)	15620
RTG	100
Nuclear Reactor	3

When incorporated into the spacecraft, an efficiency factor of 10% is incorporated into the mass of the system because going from solar to electric power or from heat to electric power incurs large losses. Examining reactors that have already been designed, these numbers are relatively accurate when looking at performances from reactors in the SNAP and SP-100 series.

A series of equations were used to find thruster mass. Since thrusters were sized very linearly based on the type and force, equations 12, 13 and Table 8 were used to calculate thruster mass.

$$M_t = a + bF \quad (12) \quad M_{Hall} = 10 \left( \frac{kg}{kW} \right) * \frac{P}{1000} \quad (13)$$

**TABLE 8: Ion Thruster mass sizing constants**

Technology	a (kg)	b (kg/mN)
Kaufman	0.93	0.037
MESC	4.2	0.045
RIT	0.36	0.083

The other subsystems were sized either as constants or as fractions of other spacecraft masses. These include: propellant tank, power conditioning unit, feed system, pressure reducing equipment and avionics.

One other issue was the clustering of ion thrusters. Thrusters have been demonstrated to be effective in clusters as large as 3. This helps systems achieve thrusts not normally achievable by a single thruster. The power increases accordingly as each thruster is added.

The design passed the criteria in the same fashion as the NTR. Total initial mass was added up and verified for under the launch limit of the Ariane 5.  $\Delta V$  was calculated with equation 9, as before. The results of the NEP system design are shown in Table 9.

**TABLE 9: NEP System Design Results**

	$\Delta V$ (km/s)	f-inert	Initial Mass (kg)	TOF (years)	Power (kWe)	# of thrusters
Jupiter (Kaufman)	15.860	0.5266	4068.58	4.13	10.258	2
Jupiter (MESC)	14.051	0.5685	3673.06	4.13	8.425	2
Jupiter (RIT)	15.433	0.5622	3768.34	4.13	9.555	2
Jupiter (Hall)	12.242	0.3351	6645.87	4.18	6.180	3
Pluto (Kaufman)	42.725	0.2656	9495.62	18.79	10.258	2
Pluto (MESC)	41.420	0.2849	8079.27	19.40	8.425	2
Pluto (RIT)	44.626	0.2826	8352.61	19.19	9.555	2
Pluto (Hall)	13.771	0.3433	6719	19.02	1.471	1

## Results and Analysis

These results validate a number of ideas and disprove others. It must be said that the given  $\Delta V$ 's were low; an actual scientific mission to Jupiter and possibly Pluto would require more  $\Delta V$  to insert the spacecraft into orbit and maneuver along the way. However, the goal of this study was to investigate the feasibility of both systems, and the accuracy of the low thrust trajectory  $\Delta V$ 's was desired over a more realistic science mission  $\Delta V$ .

Looking at the results, while NTR systems were designed for both Jupiter and Pluto missions, the Pluto mission almost failed on the volume criteria. As expected, the low density of cryogenic hydrogen causes volume issues. For any other mission with even a slightly higher  $\Delta V$ , the system would be too large for conventional launch vehicles to get into space. NEP systems were designed for both Jupiter and Pluto, however, due to the long trip time given, some analysis into the lifetime of the thruster and reactor must be done before declaring these systems truly feasible.

While the NTR's designed for this study were both capable of fulfilling the mission, it is not certain that it is the right system for the mission. On the one hand, the high thrust and impulsive burn would make the system more reliable because of its short period of operation. On the other hand, the system is much less efficient than some of the electric thrusters investigated. High specific impulse electric thrusters are able to get more payload out to the same  $\Delta V$ , and show promise in obtaining even higher  $\Delta V$ 's. There are other possible missions for NTR systems, however. With double the specific impulse of the best chemical rocket and a high thrust, NTR's could make outstanding launch vehicles. They also could be used to make high speed trips to Mars and back, reducing the exposure of human explorers to the space environment.

NEP design had a different set of issues. Electric propulsion is very efficient, but the low thrust forces lead to long trip times as shown by the Debban, McConaghy and Longuski study. With trip times of over 4 years to get to Jupiter and almost 20 years to get to Pluto, issues arise surrounding the lifetime of the equipment. Currently, the longest-running electric thruster has been tested for 30,000 hours – only 3.5 years. Much more testing needs to be done before EP systems will be launched on interplanetary missions. Space reactors are even further behind. Systems are still in the developmental stages, and need to be fully tested along with the electric thrusters to verify their reliability.

## Issues

This study has shown that nuclear propulsion is feasible for sending space craft anywhere within our solar system, and that a single launch vehicle could be used to get it to earth orbit. There are many other issues that go beyond pure feasibility, however, including testing, safety and radiological hazards.

Testing nuclear systems is a long and complicated process, costing billions of dollars. Key to determining the price of nuclear testing and development, especially NEP systems, is whether or not reactors and thrusters have to be tested for the full operational lifetime. If so, NEP testing will require enormous facilities capable of monitoring a reactor for a decade or more of operation, and ensure safety in case of mishap. Funding must be approved by Congress, and because of the long time scale of the project, it is repeatedly subject to being cut as new politicians are elected and administrations change. While this study does not use nuclear reactors for electric propulsion, it is quite possible to use them in the future, especially with the high power needed to run MPD thrusters. Testing will be difficult to conduct due to the extreme cost and the large scale of the project.

Safety and radiological hazard are the perennial issues surrounding all sorts of nuclear power. A fundamental lack of understanding about the dangers and benefits of nuclear power often clouds

the judgment of those who oppose nuclear power of all forms. There are definite hazards with nuclear power, especially when it is sent into space. Significant funding must be directed to ensure safety of the system. Launching a nuclear reactor into space will require safety procedures every step of the way to ensure nuclear material is kept away from humans and the environment. A launch vehicle blowing up on the platform or an accidental atmospheric reentry could cause all sorts of problems for those on the ground near the incident.

It can be assumed that radiological hazard is proportional to the energy released by a reactor. Therefore, a comparison can be made between the safety infrastructure needed to test a high powered NTR for a short time versus a low power NEP for a long time. See equations 14 and 15, which calculate the energy released in testing the reactors for a theoretical mission to Jupiter.

$$281,230,000 \cdot W * 9.77 \text{ min} = 164.79GJ - \text{NTR} \quad (14)$$

$$10,258 \cdot W * 4.13 \text{ yrs} = 1,336.95GJ - \text{NEP} \quad (15)$$

The NEP produces roughly 10 times as much energy and radiation over time as the NTR, increasing cost and facility requirements to test the reactor.

## Conclusions

Nuclear propulsion is a viable and useful tool for the continued exploration of our solar system. NTR's provide enormous thrusts and good specific impulses suitable for Jupiter missions, as shown, or as a Mars mission or even a launch vehicle. NEP's provide an ultra-efficient way to travel great distances, and with further development could be capable of traveling beyond our solar system. However, much stands between us and nuclear powered spacecraft. Billions of dollars must be invested in testing in order to validate the reactor and thruster designs and certify them for the thousands of hours they will be required to operate. Safety and radiological hazards must be addressed so that these systems pose no risk to humans as they leave earth. Congress and the President must then approve these systems to fly, as well as fund all of the testing. It will be a long uphill battle, but mankind stands much to gain from developing nuclear propulsion technologies.

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<sup>1</sup> Humble, Ronald W., Gary N. Henry and Wiley J. Larson, 1995, *Space Propulsion Analysis and Design*, 2<sup>nd</sup> ed., McGraw-Hill, New York, NY, p. 452

<sup>2</sup> Debban, Theresa J., T. Troy McConaghy and James M. Longuski, 2002, "Design and Optimization of Low-Thrust Gravity-Assist Trajectories to Selected Planets," AIAA 2002-4729

<sup>3</sup> Zubrin, Robert M. and Tal K. Sulmeisters, 1992, "The Application of Nuclear Power and Propulsion for Space Exploration Missions," AIAA 1992-3778

<sup>4</sup> SPAD revision 3